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EFFECT OF SLIPSTREAM ROTATION IN PRODUCING ASYMMETRIC FORCES ON A FUSELAGE

By

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EFFECT OF SLIPSTREAM ROTATION IN PRODUCING ASYMMETRIC FORCES ON A FUSELAGE

By Herbert S. Ribner and Robert MacLachlan

SUMMARY

An approximate theory of the effect of slipstream rotation on the forces on a fuselage without a wing represents the slipstream rotation by the flow about a vortex alined with the longitudinal axis. This configuration gives rise to a lateral force and yawing moment in pitch or a normal force and pitching moment in yaw. The forces are proportional to angle of inclination and to slipstream rotation as measured by the ratio of propeller torque to the square of the diameter.

A wind-tunnel investigation was made primarily to check the predictions for the lateral force on a fuselage shaped as a body of revolution. The model was tested alone and in combination with each of 6 four-blade propellers of different diameter and blade angle. The measurements were made in pitch, and a lateral force was found with a magnitude of the order of the theoretical value.

For completeness, measurements were also made of the incremental pitching moment, yawing moment, normal force, rolling moment, and thrust due to propeller operation. These measurements were somewhat ambiguous in that they represented the combined effects of fuselage interference and direct propeller forces; therefore, they were not analyzed. This ambiguity is not present in the measurement of lateral force, since a propeller in pitch is known to experience no appreciable lateral force.

INTRODUCTION

A strong tendency to yaw toward the left under conditions of high thrust and low speed is one of the difficulties experienced with single-engine fighter airplanes equipped with a single-rotating propeller. (See reference 1.) This behavior is attributed in part to the effect of the rotating slipstream in striking the vertical tail. Some unpublished tests made at the Langley Memorial Aeronautical Laboratory show, however, that a substantial yawing moment and lateral

force are still obtained when the vertical and horizontal tails are removed. A part of the yawing moment is developed by the propeller because of its inclination in pitch (references 2 and 3). The larger part may be attributed to the slipstream-interference forces on the wing and fuselage.

Speculation on the origin of the interference forces on the wing and fuselage led to an idealized picture with the slipstream rotation represented by the flow about a vortex. If this vortex is bound in the fuselage and alined with the longitudinal axis the body contour will be a stream surface of the flow. Even in the absence of a wing and tail this representation leads to a yawing moment and a lateral force, both proportional to the angle of pitch and to the slipstream rotation.

A wind-tunnel investigation was made in the Langley stability tunnel primarily to check the predictions for the lateral force on a fuselage alone. Most of the measurements were made at low thrust coefficient for reasons of practicability, since the essential features of slipstream-fuselage interference may be observed without carrying the tests to the high values of thrust coefficient at which the effects are prominent on actual airplanes. The theory is first presented in some detail, and then the experiments are described and the measurements of lateral force are compared with the predictions.

SYMBOLS

The force and moment coefficients are based on the volume of the fuselage. The forces and moments are referred to a system of rectangular body axes with origin at the one-quarter-length point of the fuselage. The X-axis lies along the longitudinal axis of the fuselage and is directed forward, the Y-axis is directed to the right, and the Z-axis is directed downward. The positive sense of a force agrees with the positive sense of the force axis and the positive sense of a moment follows that of a right-hand screw progressing in the positive direction of the moment axis.

D	diameter of propeller
R	radius of propeller
r	station radius
b	section chord

B	number of blades
β	section blade angle
β_M	blade angle at 0.75R of basic 26-inch propeller
l	length of fuselage
V	volume of fuselage
Q	propeller torque
ρ	mass density of air
U	stream velocity
q	dynamic pressure $\left(\frac{1}{2}\rho U^2\right)$
a	propeller inflow factor $\left(\frac{-1+\sqrt{1+\frac{8}{\pi}\Gamma_c}}{2}\right)$
Γ	circulation
α	angle of attack
ϵ	angle of downwash
X	longitudinal force
Y	lateral force
Z	normal force (positive downward)
N	yawing moment
C_X	longitudinal-force coefficient $\left(\frac{X}{qV^{2/3}}\right)$
C_Y	lateral-force coefficient $\left(\frac{Y}{qV^{2/3}}\right)$
C_n	yawing-moment coefficient $\left(\frac{N}{qV}\right)$

C_Z normal-force coefficient $\left(\frac{Z}{qV^{2/3}}\right)$

C_m pitching-moment coefficient $\left(\frac{\text{Pitching moment}}{qV}\right)$

C_l rolling-moment coefficient $\left(\frac{\text{Rolling moment}}{qV}\right)$

T_c' effective thrust coefficient $\left(\left(\frac{X}{qV^{2/3}}\right)_{\text{propeller on}} - \left(\frac{X}{qV^{2/3}}\right)_{\text{propeller off}}\right)$

T_c thrust coefficient based on propeller diameter $\left(\frac{V^{2/3}}{2D^2} T_c'\right)$

Q_c' torque coefficient $\left(\frac{Q}{qV}\right)$

S.F.F. side-force factor

THEORY

Under average conditions the rotation in a slipstream does not differ much from the rotation about a line vortex; that is, the rotational velocity is relatively large near the center and decreases toward the outside. Exact equivalence occurs when the circulation is constant along the propeller blades, a condition of almost uniform distribution of thrust over the disk area for lightly loaded propellers. Then no vorticity will be shed along the blades, and all the vorticity will appear in a central vortex of strength Γ and B tip vortices of strength Γ/B each, and of opposite sense where B is the number of blades.

A blade element of length dr at radius r will experience a component of force in the plane of the propeller of amount $\rho U(1 + a)\frac{\Gamma}{B} dr$, where $U(1 + a)$ is the axial velocity at the propeller disk. The total propeller torque is therefore

$$Q = \int_0^{D/2} B\,\rho U(1 + a)\frac{\Gamma}{B}\,r\,dr$$

The inflow factor a is almost constant over the propeller disk for the assumed loading. The expression is thus approximately

$$Q = \frac{\rho U(1 + a)\Gamma D^2}{8} \tag{1}$$

This expression relates the strength of the central vortex Γ to the propeller torque Q and diameter D.

If the propeller were not mounted on a body the vortex would trail freely down the center of the slipstream just as the tip vortices trail freely along a helix and constitute the slipstream boundary. The fuselage or nacelle (assumed to be a body of revolution for simplicity) must, however, be a stream surface of the flow. The determination of the flow imposes a boundary value problem of an unusual kind. The B propeller vortices of total circulation Γ may be assumed either to enter the body at the spinner or to be shed from the blade roots at the spinner. In either case this vorticity must eventually leave the body and trail with the general flow. The vorticity can leave the body at stagnation points only, and the free emergent vorticity must follow streamlines. The position of the stagnation points and the shape of these streamlines are not however known in advance because they are, in general, influenced by the vorticity.

For the particular case in which the fuselage is alined with the stream direction the problem possesses a simple solution. The central vortex Γ may be considered to pass through the body along the longitudinal axis and to continue behind the body as a free vortex, also along the axis. The vortex flow thus fits the fuselage smoothly and the position of the rear stagnation point and the straight shape of the stagnation streamline are unaffected by this flow.

If the fuselage is inclined to the stream by a small angle, the rear stagnation point, in the absence of vorticity, will still be near the rear end. Now consider the central slipstream vortex Γ to be bound along the longitudinal axis as before and to emerge as a free vortex at the stagnation point and to trail along the stagnation streamline as in figure 1. The flow induced by the bound part of the vortex is in concentric circles and so fits the fuselage smoothly. The flow induced by the free part of the vortex does not quite fit the fuselage smoothly. A small additional flow will therefore take

place - it can, in principle, be calculated by potential theory - in such a way that the combined flow will fit the fuselage smoothly. If the vortex strength Γ (which is proportional to the slipstream rotation) is small, it appears that the position of the rear stagnation point will not be greatly altered by this combination of vortex flow and vortex-induced additional flow. Thus the representation in figure 1, the vortex-induced additional flow, and the flow of the sources, sinks, and doublets that make the body a stream surface of the flow in the absence of a vortex, provide an approximation to a possible flow about the inclined fuselage in the rotating slipstream.

The sources and sinks will give rise to no forces on the body. The doublets will yield the well-known unstable moment of a fuselage in pitch. The vortex-induced additional flow will be noncirculatory and cannot contribute to the total force on the body. This flow may influence the moment, but the influence will be assumed to be small compared with that of the bound vortex of figure 1. Thus the forces on the vortex system of figure 1 are all that remain. These forces should be a first approximation to the additional forces imposed on a fuselage by slipstream rotation if the described flow actually occurs. The discussion has sought to show that this flow is possible but has not proved that it is the only possible flow.

The influence of the propeller tip vortices on the flow about the fuselage has thus far been ignored. These tip vortices form a helix of which the main effect is to induce the well-known slipstream axial inflow velocity. This inflow velocity is accounted for by the factor a in the equations. A secondary effect, which disappears if the number of propeller blades is infinite, is a small influence on the rotation in the slipstream. This secondary effect may be neglected if the angle of attack of the fuselage is small so that the fuselage nowhere approaches the slipstream boundary.

If the fuselage is at an angle of attack α the bound vortex is inclined by the angle $\alpha - \epsilon$ to the local stream velocity, where ϵ is the downwash produced by the propeller. (See fig. 1.) The fuselage should therefore experience a lateral force

$$Y = -\rho U \Gamma l (1 + 2a) \sin (\alpha - \epsilon) \qquad (2)$$

If the propeller has the usual right-hand rotation (viewed from the rear), Γ is positive and the lateral force is negative (toward the left). The center of pressure is at the center of the fuselage; there should be, therefore, a yawing moment about the quarter-length point of the fuselage of amount

$$N = \frac{\rho U \Gamma l^2}{4} (1 + 2a) \sin (\alpha - \epsilon) \qquad (3)$$

For right-hand rotation this moment is positive.

Corresponding expressions result for a normal force and pitching moment due to yaw. For right-hand rotation the normal force should be upward (negative) and the pitching moment negative.

The values of lateral force and yawing moment given by equations (2) and (3), respectively, are a little too large. A small opposing lateral force induced at the rear end of the fuselage by the curved part of the trailing vortex has been neglected. Also, the helical tip vortices are deflected downward somewhat by the flow about the fuselage. This deflection induces a small further reduction in lateral force. The exact lateral force can be obtained by a consideration of the lateral momentum associated with the relative displacement of the trailing central vortex and the helix center line far back in the wake.

The circulation of an actual propeller blade will not be constant as assumed in the simple theory. The reduction of blade width near the shanks and the departure of the shanks from airfoil shape will cause an appreciable amount of right-hand vorticity to be shed outside the fuselage. This part of the vorticity will trail with the general flow. Some attempts have been made to evaluate, by crude approximations, the influence of this free-trailing vorticity. The results suggest that equations (2) and (3) still provide solutions of the right order of magnitude. A really quantitative solution would present formidable difficulties.

A more pictorial interpretation may be made of the origin of these forces. The rotating slipstream is considered to be constrained to follow the fuselage so that the axis of rotation is approximately alined with the longitudinal axis. If the fuselage is at an angle of attack and the rotation in the slipstream has a right-hand sense, the rotational velocity has a downstream component to the left of the fuselage and an upstream component to the right of the fuselage. The resultant velocity is therefore greater on the left side of the fuselage than on the right side. According to Bernoulli's principle the pressure on the left side must be less than the pressure on the right side. The result is a lateral force to the left.

The addition of a wing to the fuselage would appear to have two effects. First, the wing would remove a large part of the slipstream rotation. Second, the downwash of the wing in pitch would considerably reduce the effective inclination of the rear part of the fuselage. The theory may therefore be extended to the case in which a wing is present by appropriately reducing the strength of the central slipstream vortex behind the center of pressure of the wing and by taking

into account the downwash behind the wing in computing the forces on the central slipstream vortex. Addition of the wing has thus moved the center of pressure of the lateral force far forward. The resultant yawing moment is toward the left for conventional cases, whereas for the fuselage alone it is toward the right.

In the tests to be described the downwash angle ϵ is small and may be taken, with sufficient accuracy, equal to the theoretical downwash far behind the isolated propeller. This downwash is associated with the propeller normal force and can be determined by the procedures of reference 3. The computed values of $1 - \frac{d\epsilon}{d\alpha}$ at zero thrust for the six propellers that were tested are included in table I. The values for zero thrust are considered representative because most of the tests were run at low values of T_c. For the same reason the propeller inflow factor a may be disregarded in comparing the tests with theory.

APPARATUS AND MODELS

The experimental investigation was conducted in the 6- by 6-foot test section of the Langley stability tunnel. The model was mounted on a single strut extending from the rear of the fuselage to the tunnel balances. The strut was constructed of hollow steel tubing and served also to house all the motor leads. The arrangement is shown in figure 2.

The model consisted of a fuselage of circular cross section which was tested alone and in combination with each of six different propellers. The fuselage was made of mahogany to the dimensions given in the table contained in figure 3. Figure 3 shows also the fuselage-support junction and the location of the propellers.

Six four-blade right-hand wooden propellers were tested. (See fig. 4.) The propellers were formed from identical 26-inch propellers by cutting off the ends of four to provide diameters of 12 inches, 19 inches, and 26 inches. Three of the propellers had blade angles of 18.9° at 0.75 of a 13-inch radius, and the remaining three had blade angles of 39.4° at the same station. The blade-form curves for the two basic 26-inch propellers are given in figure 5. Side-force factors (reference 4) for the six propellers, estimated from these curves, are given in table I. The values for the 26-inch propellers were obtained by the method of reference 3 and the values for the cut propellers were obtained from these values by the considerations of reference 4. The blade sections inboard of 5.2 inches

from the center were assumed to make no contribution to the side-force factor.

The propellers were driven by a three-phase induction motor for which a torque-calibration curve was available. The rotational speed was controlled by varying the frequency of the current input.

TESTS

The fuselage alone and in combination with each of six propellers was tested through an angle-of-attack range from -1° to 30°. The angle of yaw was zero at all times. The propeller torque was held constant during each run. The values of torque used in the tests ranged from 1.6 to 8 foot-pounds in increments of 1.6 foot-pounds. For the higher-pitch propellers of 26-, 19-, and 12-inch diameter the upper limits of torque were 3.2, 4.8, and 8 foot-pounds, respectively; for the lower-pitch propellers of 26-, 19-, and 12-inch diameter, the upper limits of torque were 4.8, 6.4, and 8 foot-pounds, respectively.

All tests were made at a dynamic pressure of 19.9 pounds per square foot, which corresponds to a velocity of about 91 miles per hour. The Reynolds number based on the total length of the fuselage was about 3,000,000.

PRESENTATION OF DATA

The data have been corrected for deflection of the model support under load. In the absence of a suitable theory no corrections have been applied for the effect of the tunnel-wall constraint.

The variation with angle of attack of the force and moment coefficients for the fuselage alone is presented in figure 6. These values were subtracted from the corresponding values for the propeller-fuselage combinations and the net results are presented as increments in figure 7. The coefficients plotted in figure 7 therefore represent the forces and moments acting on an isolated propeller plus the additional forces and moments on both propeller and fuselage due to propeller-fuselage interference.

RESULTS AND DISCUSSION

A lateral force and yawing moment due to slipstream rotation in conjunction with pitch appear in the experimental results of figure 7, as predicted by theory. The prediction for lateral force (equations (1) and (2)) may be expressed in the coefficient form

$$\frac{C_Y D^2}{Q_c' V^{2/3}} = \text{Constant} \times \alpha \qquad (4)$$

if the slipstream factor a is neglected as being small. The values of the theoretical constant for the six propellers are included in table II. The experimental values of $\frac{C_Y D^2}{Q_c' V^{2/3}}$ for Q_c' up to 0.38 are plotted in figure 8 with the theoretical line (equation (4)) for comparison. The agreement at small angles of attack varies from poor to good. The average slope through zero for the 23 experimental curves is about 15 percent less than the theoretical slope and the average individual scatter is $\pm$11 percent. (Individual values are given in table II.) The simple theory of propeller-fuselage interference thus provides a fairly quantitative prediction of the lateral force due to pitch or normal force due to yaw.

The variation of lateral-force coefficient C_Y with torque coefficient Q_c' is shown in figure 9 for fixed diameter and angle of attack. The variation shows the theoretical linearity only up to $Q_c' \approx 0.38$. The theory implies small values of Γ, or correspondingly small values of Q_c'; therefore measurements at higher values of torque coefficient were not used in the preparation of figure 8.

Quantitative comparison could not be made with the theoretical value of the fuselage yawing moment because the measurements included a propeller yawing moment (references 2 and 3) of uncertain magnitude. The sense of the combined moments at small angles of attack agrees however, with the theoretical prediction. Four other quantities - pitching moment, normal force, rolling moment, and thrust - are included for completeness in figure 7. These measurements, like the measurements of yawing moment, are somewhat ambiguous in that they represent the combined effects of fuselage interference and direct propeller forces; therefore they were not analyzed. This ambiguity, however, is not present in the measurement of the lateral force, since a propeller in pitch is known to experience no appreciable lateral force.

CONCLUDING REMARKS

Slipstream rotation about a fuselage without wing or tail leads to a fuselage lateral force and yawing moment for pitch or a fuselage normal force and pitching moment for yaw. The forces are proportional to slipstream rotation, measured by the ratio of propeller torque to the square of the diameter, and to angle of inclination.

These forces are predicted by an approximate theory that represents the effect of slipstream rotation by the flow about an equivalent vortex bound in the fuselage and alined with the longitudinal axis. Wind-tunnel measurements in pitch for 6 four-blade propellers of several diameters and blade angles yielded values of fuselage lateral force of the order of magnitude predicted by the theory.

The theoretical representation may be extended to the case of a fuselage with a wing by reducing the vortex strength behind the center of pressure of the wing to allow for the slipstream rotation removed by the wing and by taking into account the downwash from the wing. A left yawing moment caused by pitch is indicated for the fuselage with wing in contradistinction to the right yawing moment found for the fuselage without wing.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., November 15, 1946

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1. Phillips, William H., and Vensel, Joseph R.: Measurements of the Flying Qualities of a Supermarine Spitfire VA Airplane. NACA ACR, Sept. 1942.

2. Harris, R. G.: Forces on a Propeller Due to Sideslip. R. & M. No. 427, British A.C.A., 1918.

3. Ribner, Herbert S.: Notes on the Propeller and Slipstream in Relation to Stability. NACA ARR No. L4112a, 1944.

4. Ribner, Herbert S.: Proposal for a Propeller Side-Force Factor. NACA RB No. 3L02, 1943.

TABLE I

SIDE-FORCE FACTORS FOR THE SIX PROPELLERS AND DOWNWASH ASSOCIATED WITH THE NORMAL FORCE

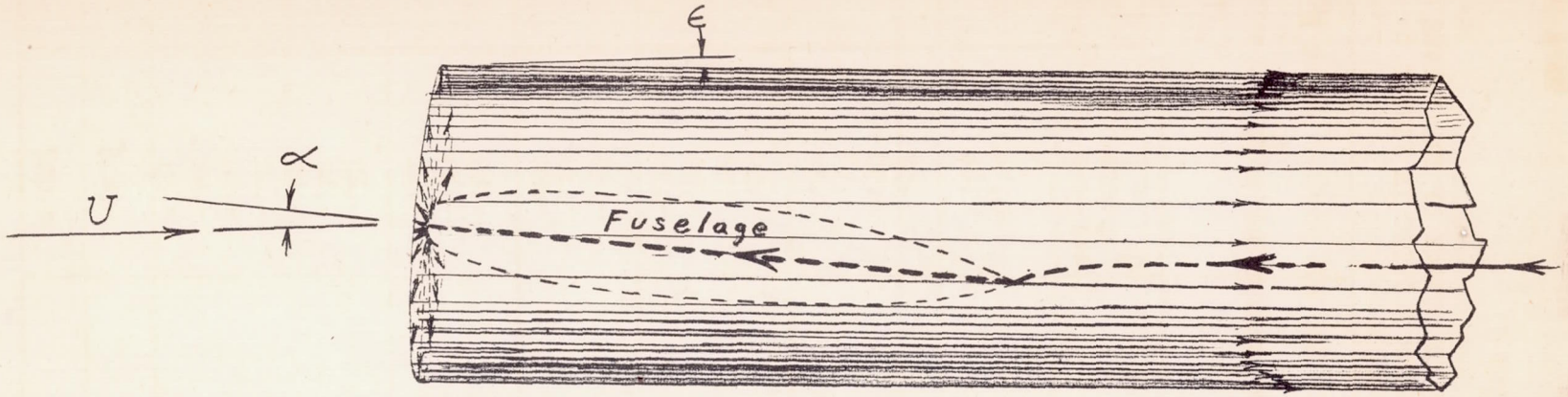
β_M (deg)	D (in.)	S.F.F.	$1 - \frac{d\epsilon}{d\alpha}$
39.4	26	101.4	0.918
	19	159.0	.872
	12	234.0	.811
18.9	26	99.0	.949
	19	162.0	.917
	12	254.0	.870

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TABLE II

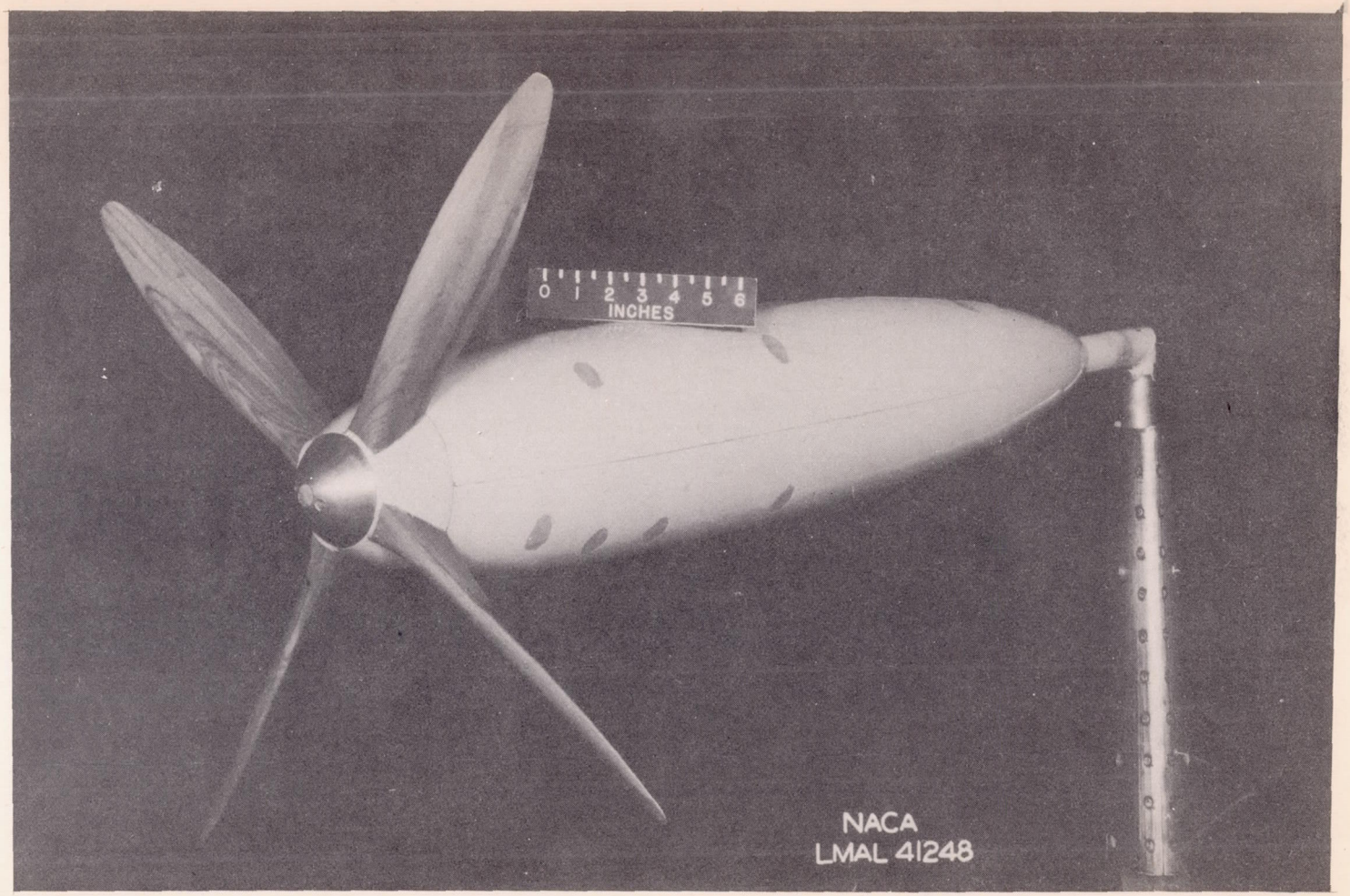
SLOPE THROUGH ZERO ANGLE OF ATTACK OF THE CURVES OF LATERAL-FORCE PARAMETER PLOTTED IN FIGURE 8

β_M (deg)	D (in.)	Q_c'	$\frac{d}{d\alpha}\left(\frac{C_Y D^2}{Q_c' V^{2/3}}\right)_{\alpha=0}$	
			Experimental	Theoretical
39.4	26	0.13 .25	-0.44 -.44	-0.51
	19	.13 .17 .25 .38	-.57 -.55 -.43 -.37	-.48
	12	.06 .13 .25 .38	-.59 -.42 -.40 -.39	-.45
18.9	26	.13 .25 .38	-.35 -.33 -.29	-.53
	19	.13 .25 .38	-.40 -.35 -.31	-.51
	12	.06 .10 .12 .13 .21 .25 .38	-.42 -.47 -.42 -.42 -.40 -.41 -.39	-.48
Average			-.416	-.49
Average Scatter			±.052	———



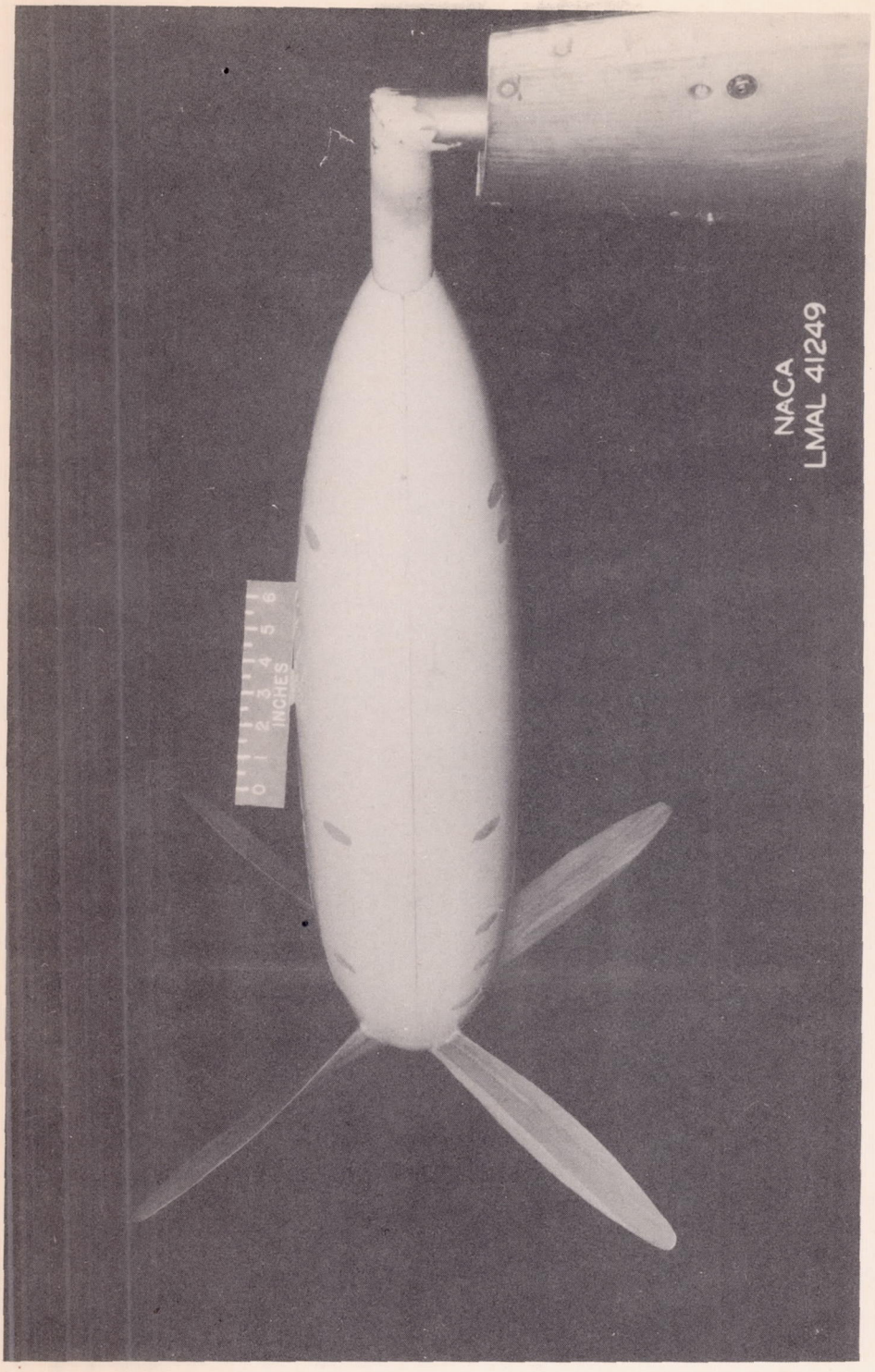
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Figure 1.— Suggested approximate model for propeller-fuselage interference. (For simplicity only the longitudinal components of the vortices in the slipstream boundary are shown.)



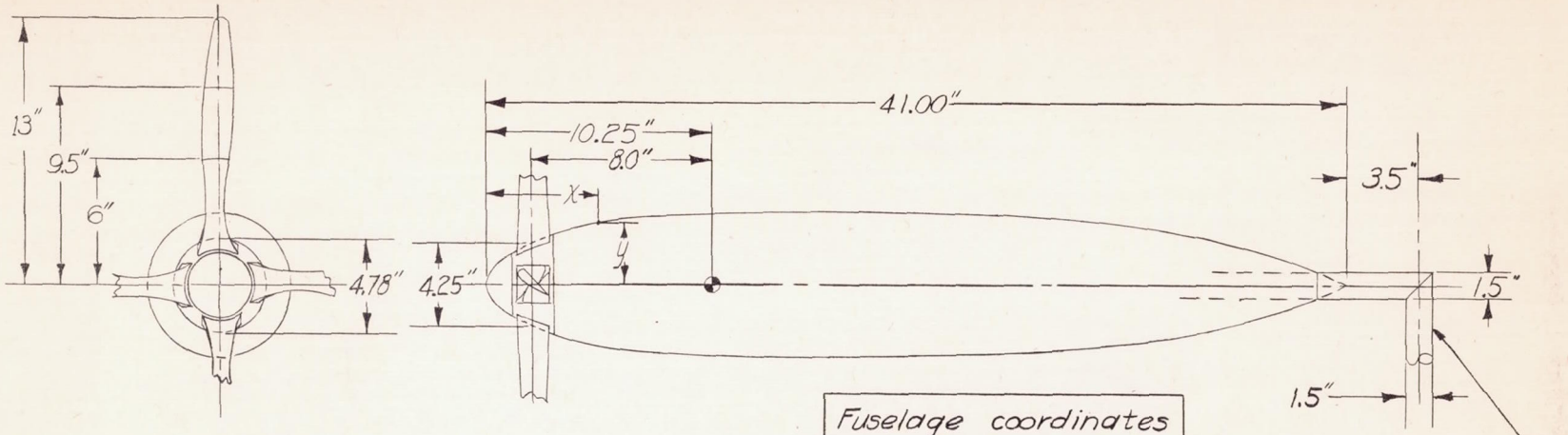
(a) Three-quarter front view.

Figure 2.- Propeller-fuselage-interference model with 26-inch-diameter propeller.



(b) Three-quarter rear view.

Figure 2.- Concluded.



Fuselage volume = 0.634 cu ft

Fuselage coordinates			
x	y	x	y
.000	.000	20.659	3.464
.318	.786	24.727	3.323
.827	1.264	28.796	3.041
1.335	1.599	32.864	2.559
2.352	2.079	34.898	2.207
4.387	2.695	36.932	1.726
8.455	3.293	38.966	1.017
12.553	3.468	39.983	.557
16.591	3.499	41.000	.000

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Figure 3. - Propeller-fuselage-interference model.

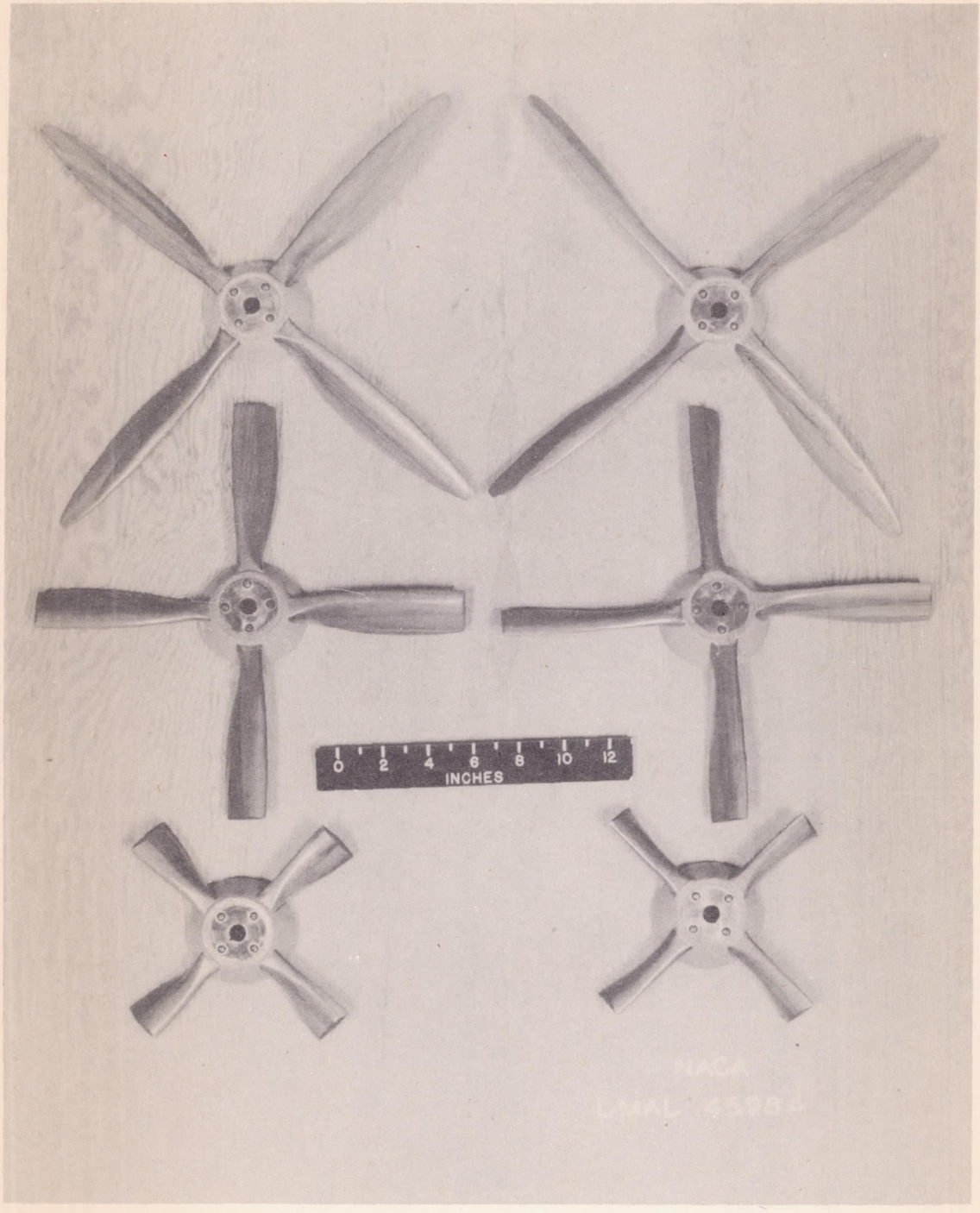


Figure 4.- Six propellers used for propeller-fuselage-interference tests.

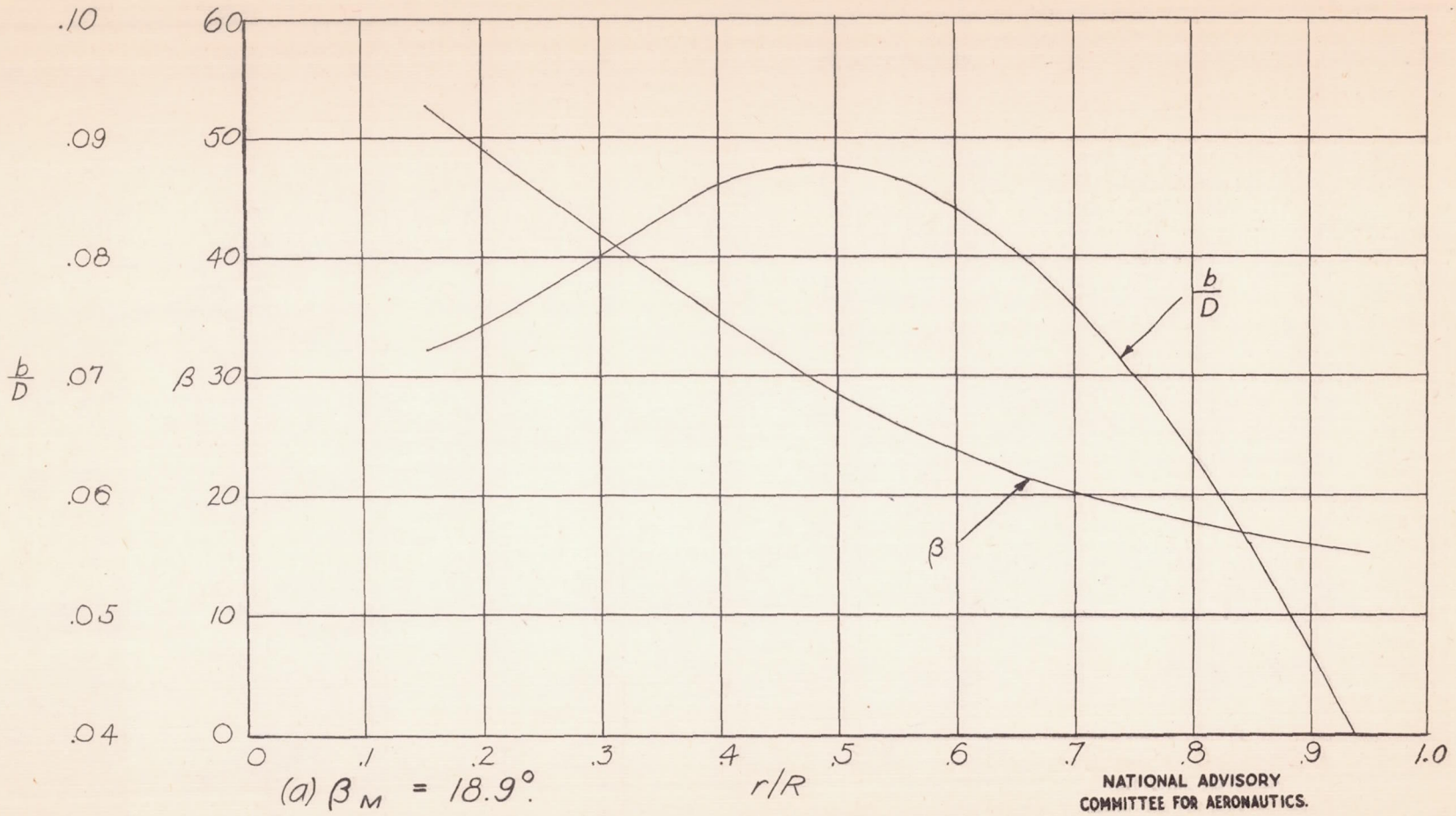
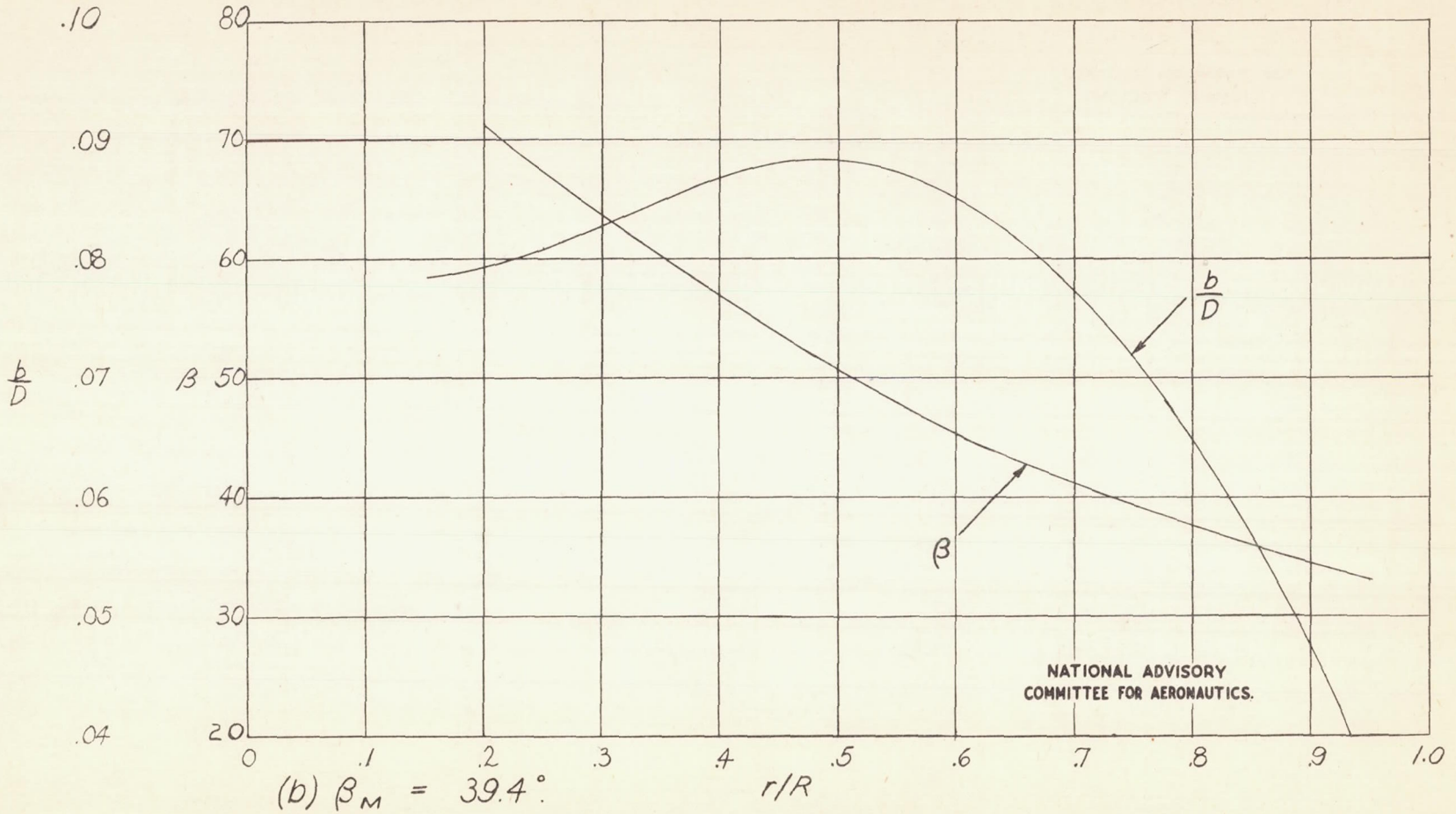


Figure 5.- Plan forms and pitch distributions of basic 26-inch-diameter propellers of approximate Clark Y section.



(b) $\beta_M = 39.4°$.

Figure 5.- Concluded.

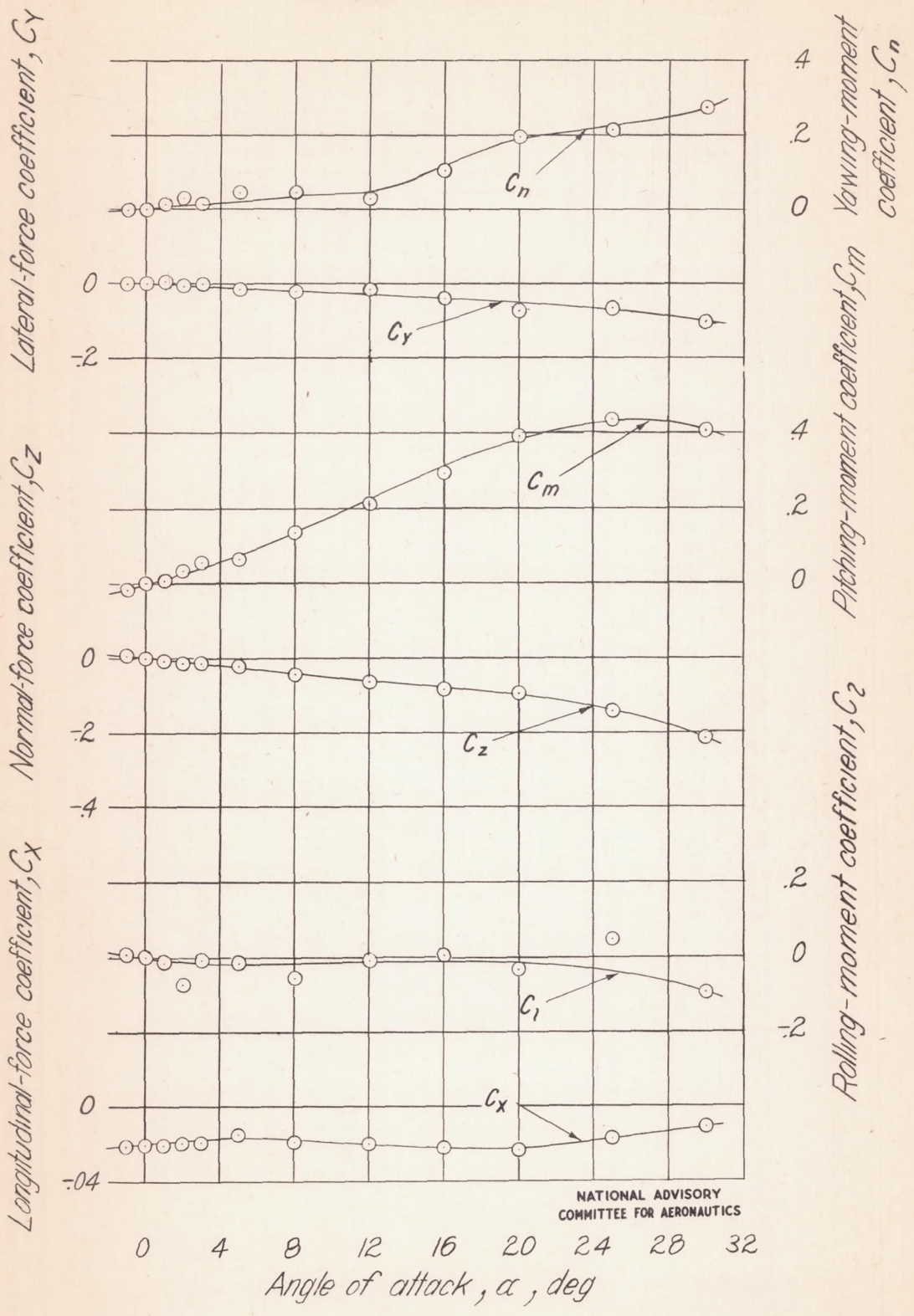
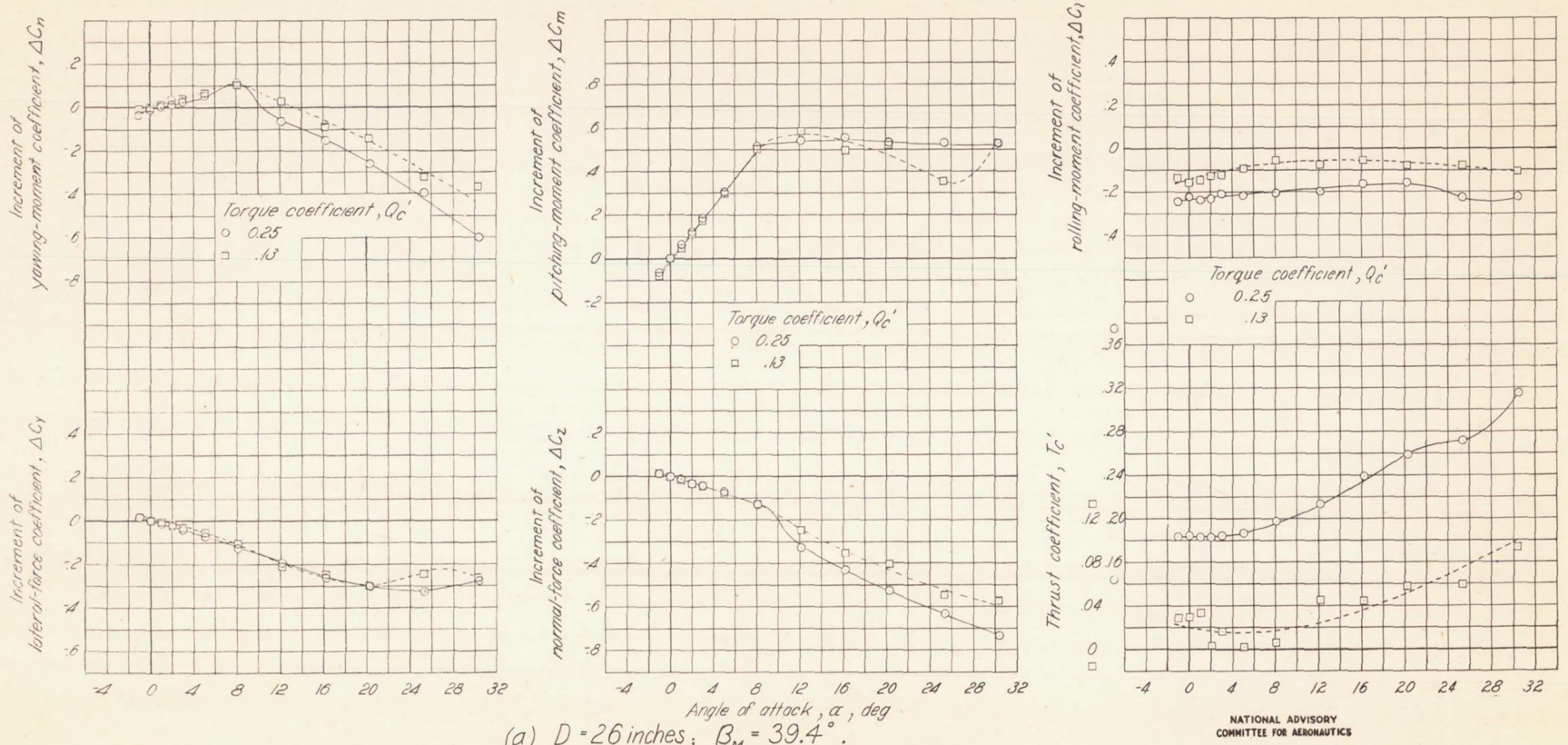
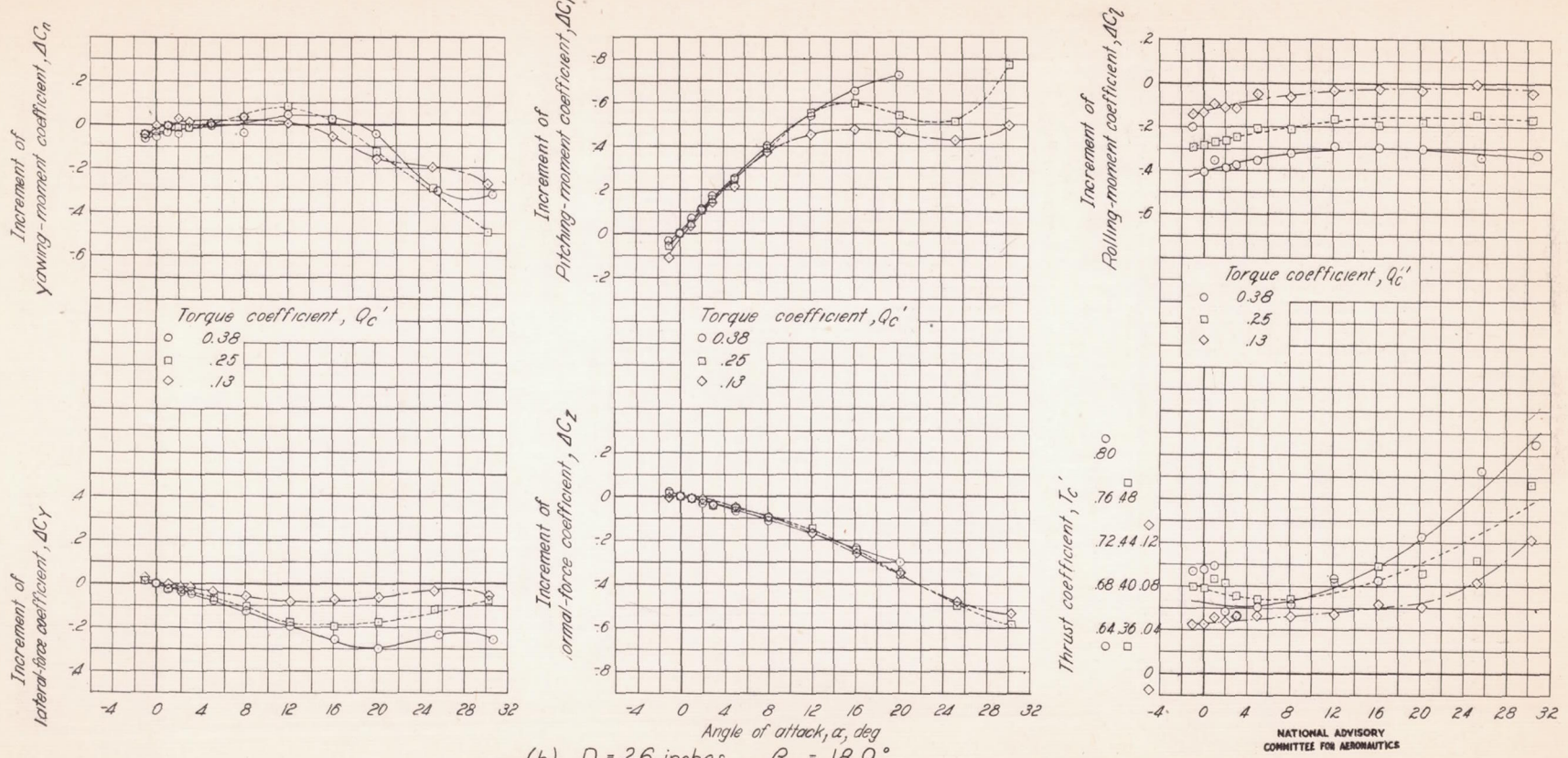


Figure 6.-Aerodynamic characteristics of model with propeller off.



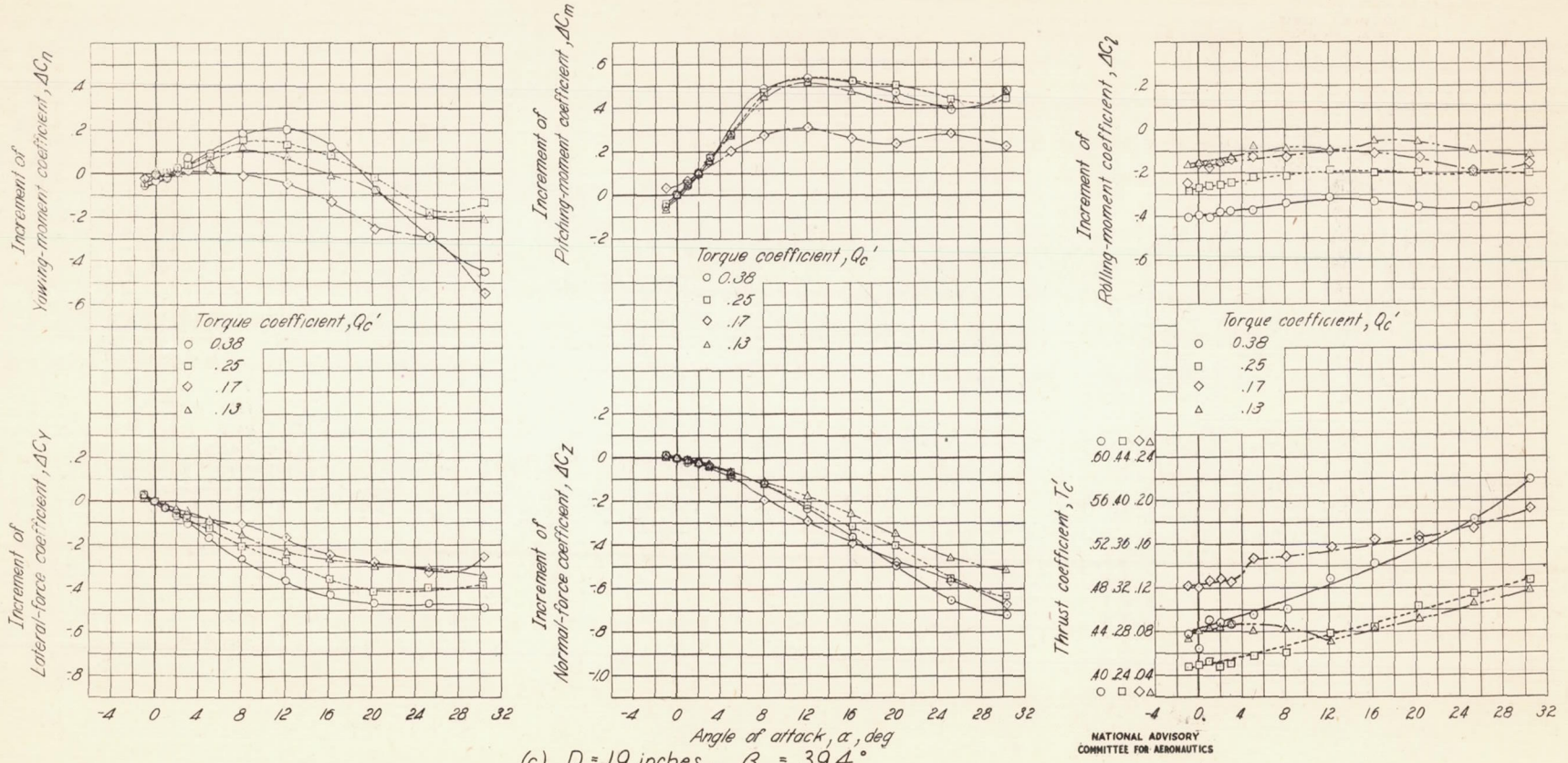
(a) D = 26 inches; β_M = 39.4°.

Figure 7. – Increments of coefficients of forces and moments on the propeller and fuselage caused by propeller rotation.



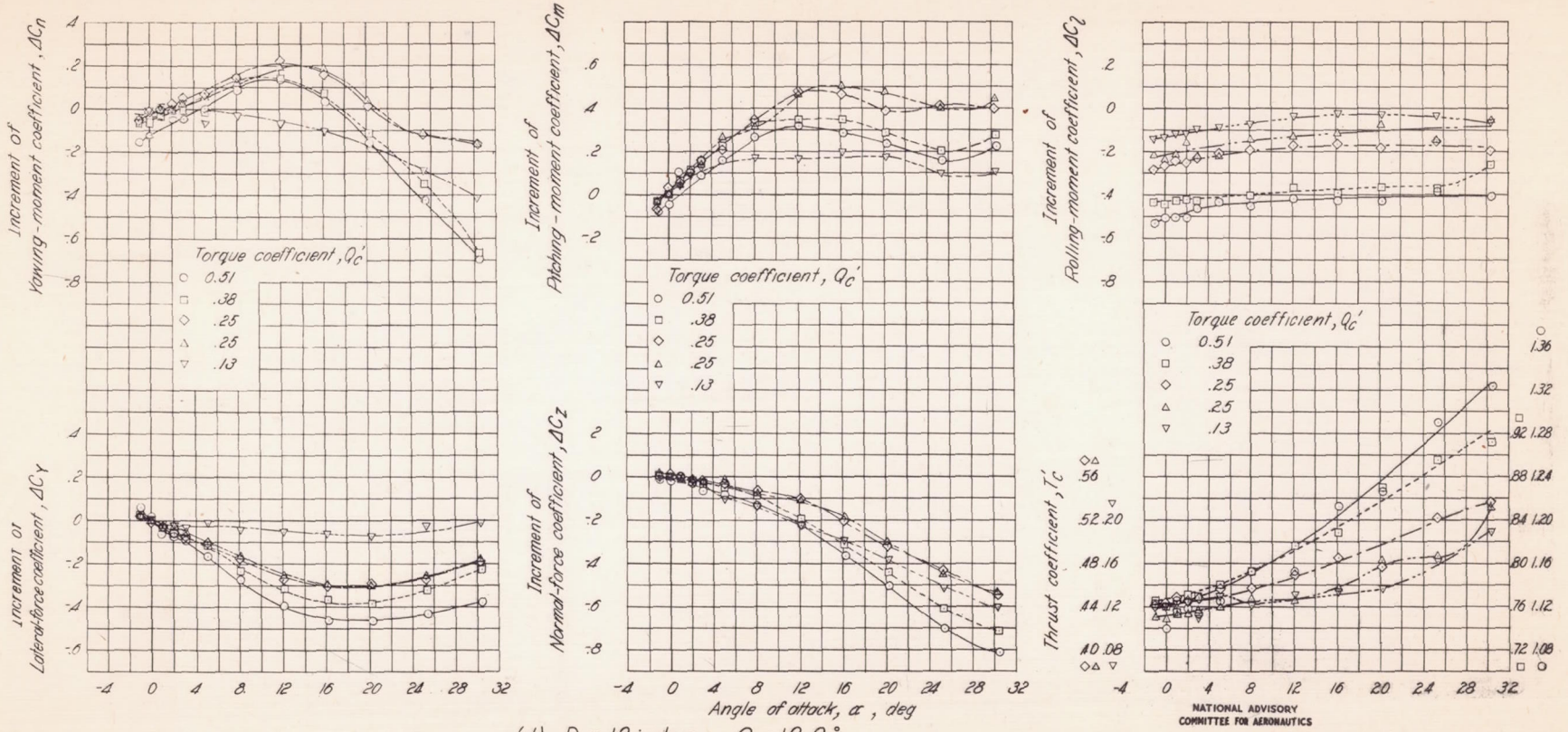
(b) $D = 26$ inches; $\beta_M = 18.9°$.

Figure 7. - Continued.



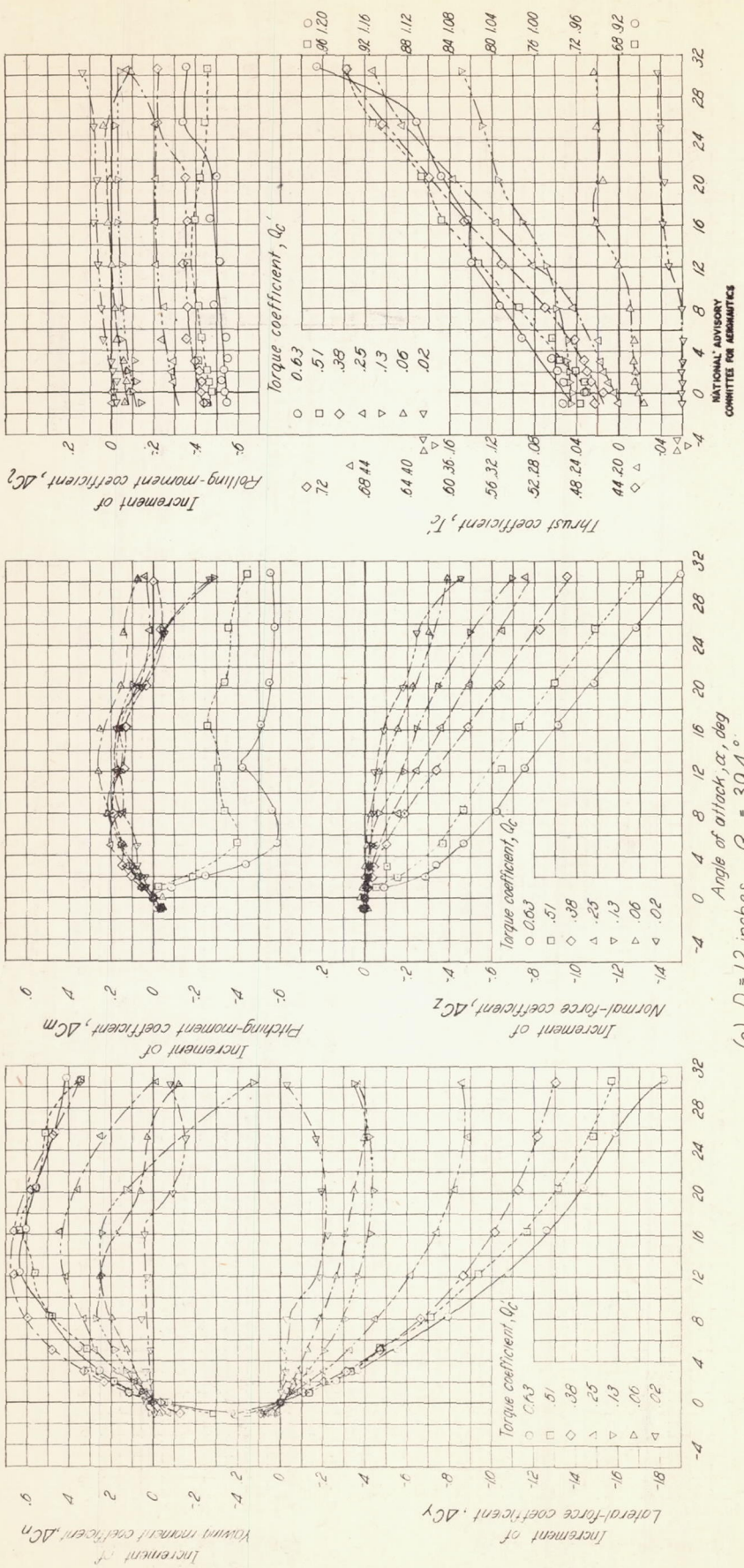
(c) D = 19 inches; β_M = 39.4°.

Figure 7.- Continued.



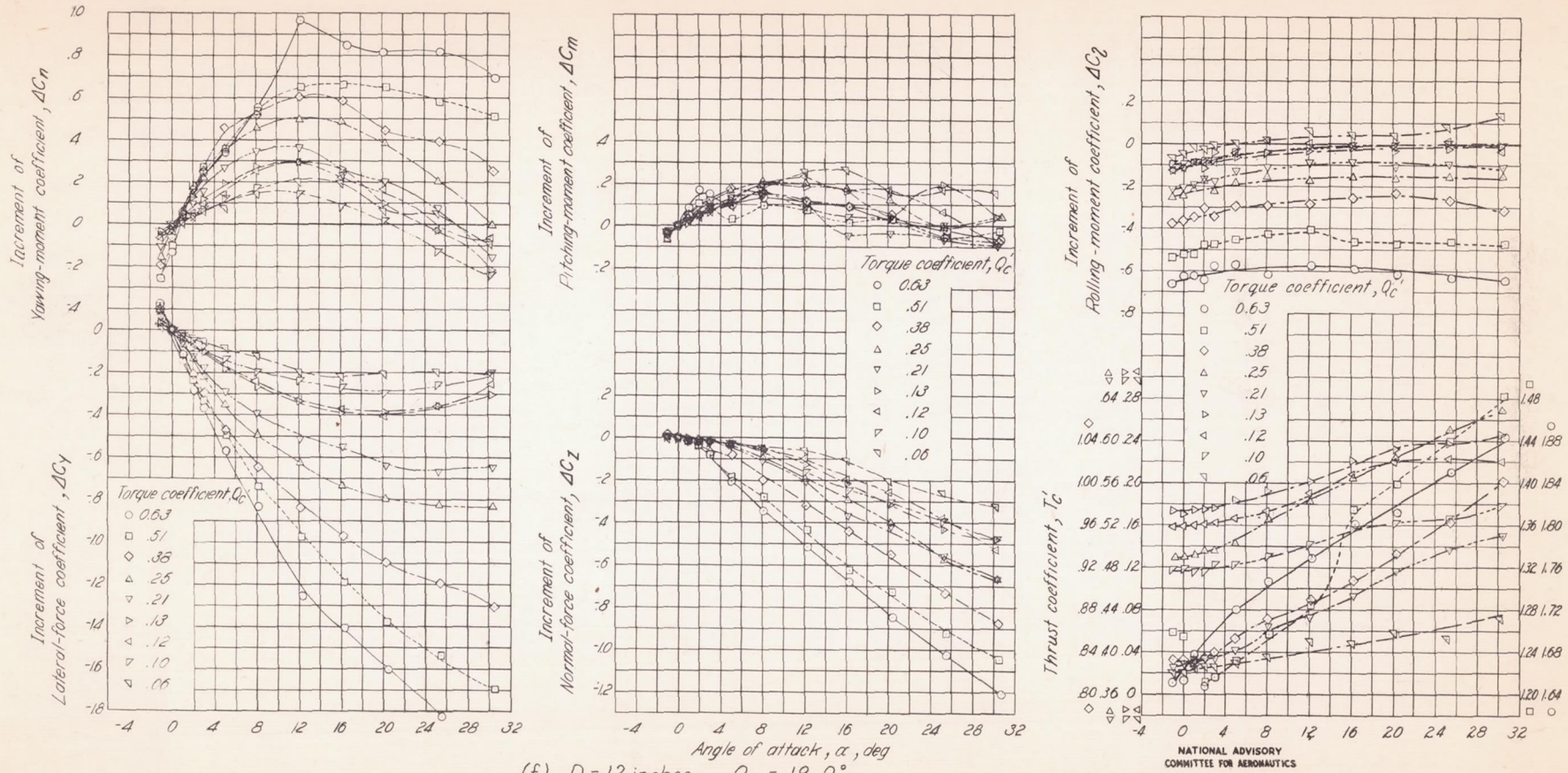
(d) D = 19 inches; $\beta_M = 18.9°$.

Figure 7. - Continued.



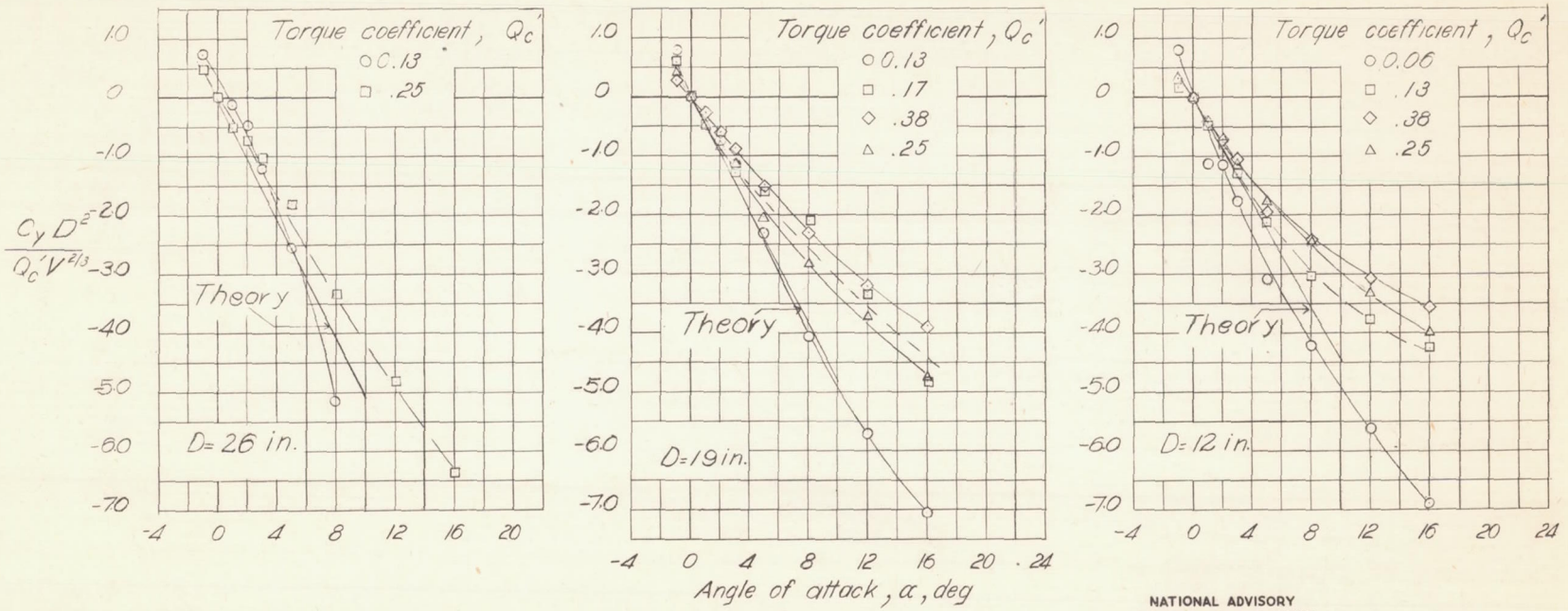
(e) $D = 12$ inches; $\beta_M = 39.4°$.

Figure 7. - Continued.



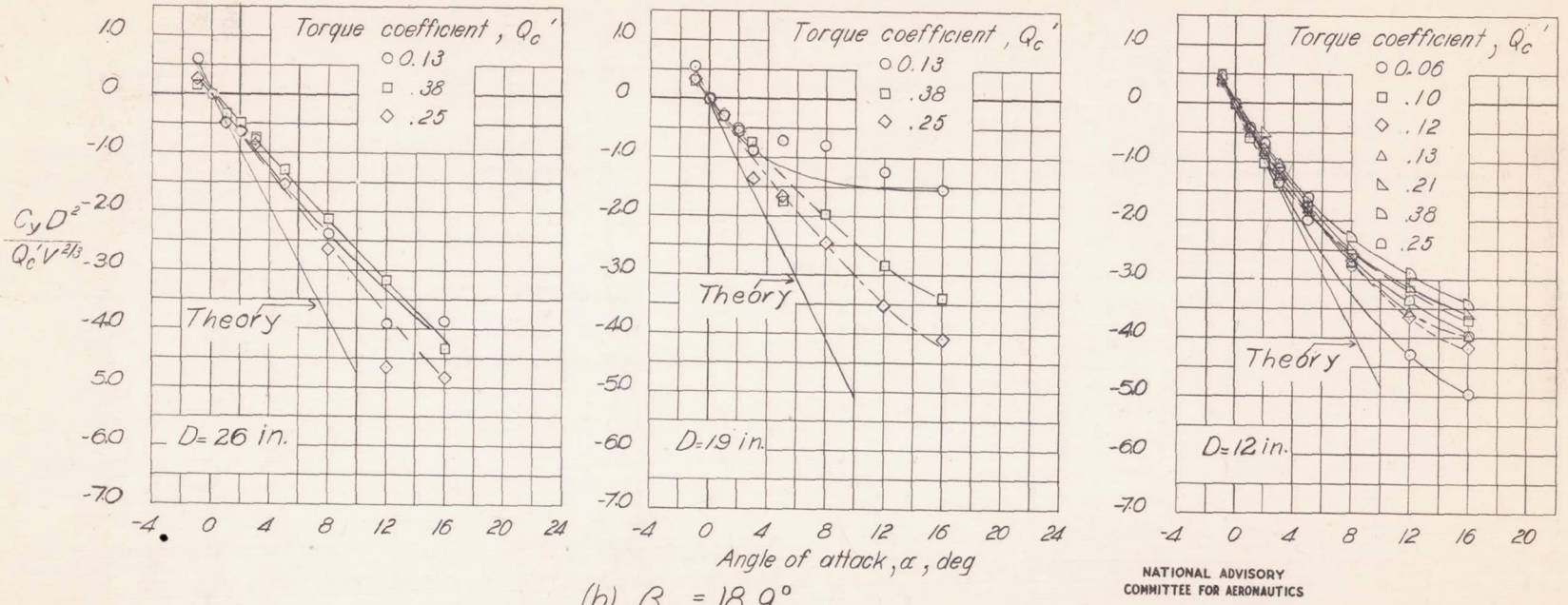
(f) D = 12 inches; $\beta_M = 18.9°$.

Figure 7. - Concluded.



(a) $\beta_M = 39.4°$.

Figure 8. - Variation of $C_y D^2/Q_c' V^{\frac{2}{3}}$ with angle of attack.



(b) $\beta_M = 18.9°$.

Figure 8.- Concluded.

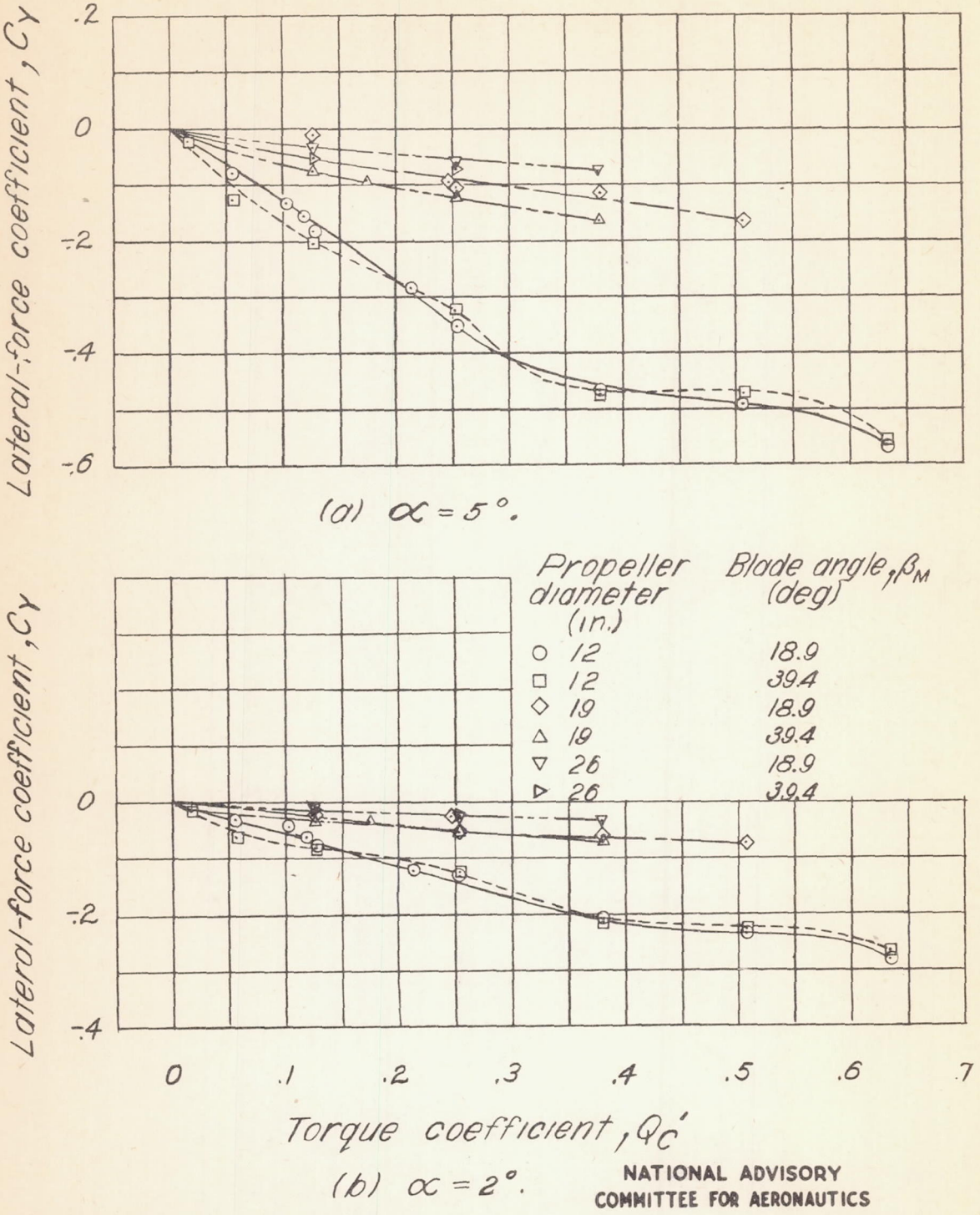


Figure 9.- Variation with torque coefficient of lateral-force coefficient resulting from propeller-fuselage interference.